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RESEARCH MEMORANDUM

A FLIGHT INVESTIGATION OF THE EFFECT OF LEADING-EDGE CAMBER ON THE AERODYNAMIC CHARACTERISTICS OF A SWEPT-WING AIRPLANE

By Seth B. Anderson, Frederick H. Matteson,
and Rudolph D. Van Dyke, Jr.

Ames Aeronautical Laboratory
Moffett Field, Calif.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

Flight measurements were made on a swept-wing jet aircraft to determine the effects of adding forward camber and an increased leading-edge radius on the low-speed stalling characteristics, the high-speed static longitudinal stability, and the airplane drag.

The results showed that the modified leading edge produced values of maximum lift somewhat greater than that given by the slats on the normal airplane; however, the stall was unacceptable because of an abrupt roll-off. The addition of a fence resulted in a satisfactory stall with values of maximum lift comparable to the normal airplane. The modified leading edge produced no significant changes in the longitudinal-stability characteristics in the transonic Mach number range. The drag of the airplane with the modified leading edge was slightly higher than that of the normal airplane below 0.86 Mach number and above 0.94 Mach number at a normal force coefficient of 0.15.

INTRODUCTION

High-lift devices such as leading-edge slats and leading-edge flaps have been used successfully to delay flow separation and thereby improve the low-speed lift characteristics of swept wings. These devices, however, are mechanically complicated, add appreciable weight to the wing structure, render useless the forward portion of the wing for fuel storage, and complicate the installation of de-icing equipment. Reference 1 gives results of wind-tunnel tests of an airplane with a modified wing section, incorporating a moderate amount of camber over the forward portion of the chord and an increase in leading-edge radius which serve

to delay flow separation to at least as high lift coefficients as are attainable with a slotted wing.

The results of that investigation, conducted on a full-scale 35° swept-wing airplane in the Ames 40- by 80-foot wind tunnel, left open three questions regarding the use of the modified leading edge in actual flight. One was the effect on the low-speed stalling characteristics since the modified leading edge produced a sharp lift-curve top and longitudinal instability beyond maximum lift; second was the effect on longitudinal stability at supercritical speeds; and third was the effect on the airplane high-speed drag.

In order to answer these questions, a flight investigation was carried out on the same type of swept-wing airplane described in reference 1. The results of the flight investigation are reported herein.

NOTATION

A_Z ratio of net aerodynamic force (positive when directed upward) along airplane Z axis to the weight of airplane

C_D drag coefficient, drag/qS

C_L lift coefficient, lift/qS

C_m pitching-moment coefficient, pitching moment/$qS\bar{c}$

C_{m_o} pitching-moment coefficient at zero lift

$C_{m_{w+f}}$ wing-fuselage pitching-moment coefficient

C_N normal-force coefficient, normal force/qS

M Mach number

$\bar{c}$ mean aerodynamic chord of wing, ft

c local wing chord, ft

q dynamic pressure, $\frac{1}{2}\rho V^2$, lb/sq ft

S wing area, sq ft

V true airspeed, ft/sec

ρ density, slugs/cu ft

α airplane angle of attack, deg

α_{L0} angle of zero lift, deg

δ_e elevator angle, deg

EQUIPMENT AND TESTS

The test airplane, which was the same type studied in reference 1, was a jet-powered fighter having swept-back wing and tail surfaces. A photograph of the airplane is presented in figure 1 and a two-view drawing is given in figure 2. A description of the geometric details of the normal airplane is given in table I. Figure 3(a) is a drawing showing the unmodified and modified wing airfoil sections at station 0.857 semispan. The wing with this modified section is the same as that of modification 1 of reference 1. Figure 3(b) is a photograph showing the leading-edge modification. Geometric details of the modified wing airfoil section are contained in reference 1 (listed as modification 1). This leading-edge modification tested in flight extended over the complete span of the wing and was made of wood. Figure 4 shows a view of a 0.25c fence at 0.63 semispan. This fence was a wrap-around type, approximately 5 inches high. The fence was on the wing only during the stall tests of the modified leading edge.

Standard NACA instruments and an 18-channel oscillograph were used to record the various quantities. Airspeed measurements and values of Mach number were obtained using the nose-boom airspeed system described in reference 2. For the stall flights a free-swiveling airspeed head was used to minimize the error caused by angle of attack. Horizontal-tail loads used to derive the wing-fuselage pitching moments were measured by means of strain gages at the three pin-joined attachment fittings where the tail is joined to the fuselage. Angle of attack was measured by a vane mounted 8 feet ahead of the fuselage nose.

All coefficients are based on the dimensions of the unmodified airplane.

Flight tests to measure the low-speed stalling characteristics were made at 10,000 feet altitude. All other testing was performed at approximately 35,000 feet.

RESULTS AND DISCUSSION

Low-Speed Stalls

Maximum lift characteristics.- The lift curves determined in flight for the test airplane with the modified leading edge, the basic wing (slats locked closed and sealed), and the normal slat configuration are given in figure 5. It will be noted that for the flaps-down case the modified leading edge provided lift coefficient increments approximately 0.31 greater than that of the basic wing and 0.22 greater than that with the slats operating. These increments corresponded to those indicated by the tunnel tests, although the absolute value of maximum lift was higher in the tunnel for the flaps-down case. The maximum lift occurred at a higher angle of attack for the modified leading edge than for the normal wing with slats operating for both the flaps-up and flaps-down conditions.

Nature of stall.- The stalling characteristics of the airplane with the modified leading edge for the flaps-up and flaps-down conditions were considered unacceptable because of an abrupt roll-off and the large angles of bank attained at the stall. In addition, the stall was made more hazardous by the absence of any stall warning. The records showed that the initial roll-off at the stall resulted in an angle of bank of approximately 60° and a rolling velocity of the order of 1.4 radians per second. The magnitude of these quantities increased in value rapidly (in excess of the ranges of the instruments) as the stick was brought back. Similar unacceptable characteristics existed for the airplane with the basic wing (slats locked closed and sealed). For the normal airplane (slats operating) the stall was considered operationally satisfactory. In this case, the stall was characterized by a more gradual departure from wings-level flight with the initial angle of bank being less than 5° and the rolling oscillations building up to a maximum amplitude of about $\pm 50^\circ$ in 10 seconds of stalled flight. It is of interest to note that the pilot did not notice a pitch-up beyond maximum lift with the modified leading edge even though the wind-tunnel results presented in figure 6 (taken from ref. 1) showed a marked unstable break in pitching moment. The pilot did note the small region of neutral stability for the flaps-up case (fig. 6) immediately before maximum lift and a pitch-down beyond maximum lift for the normal airplane with the slats operating.

Observations of tufts on the upper wing surface at the stall indicated that the abrupt roll-off for the modified leading edge was due to an asymmetric flow separation initiating outboard near the wing tip and spreading inboard rapidly. This abrupt stall was also evidenced by the sharp lift peaks measured in flight and in the wind tunnel (fig. 5). In an effort to improve the stalling characteristics, a number of wing

modifications aimed at flattening out the lift peak and thus modifying the abruptness of the stall were flight-tested. The modifications suggested in the wind-tunnel results of reference 1 (wing modifications 2 and 3), which were designed to spoil the flow at the wing leading edge at the root, did result in some improvement but the stall characteristics for the flaps-down case were still considered unsatisfactory. Tuft pictures taken in flight for the flaps-down case showed that modification 3 produced areas of separation inboard; however, separation still occurred rather abruptly over the outboard wing panel. This modification resulted in a decrement in $C_{L_{max}}$ of 0.31 for flaps up and 0.21 for flaps down. These decrements were of the same order of magnitude as those of the wind-tunnel results.

It had been noted from previous tests on the basic wing that the use of fences had provided satisfactory stalling characteristics with some penalty in maximum lift. A number of flight tests were made with fences on the wing with the modified leading edge in an effort to provide satisfactory stalling characteristics with a minimum penalty in maximum lift. The optimum configuration found was that shown in the photograph of figure 4, consisting of a short chord fence (0.25c) at 0.63 semispan. This fence produced stalling characteristics similar to the normal wing with slats operating, that is, a gradual departure from wings-level flight with the initial angle of bank less than 5° and a build-up in rolling oscillations to about ±50° over a period of 10 seconds of stalled flight. The stall warning for this configuration was substantially the same as that for the normal airplane, that is, satisfactory with flaps up and marginally satisfactory to unsatisfactory with flaps down. The stall warning for the normal airplane was considered satisfactory with flaps up and unsatisfactory with flaps down. The decrement in maximum lift for the fence installation is of the order of 0.17. (See fig. 7.) Thus the maximum lift of the airplane with cambered leading edge and fence installed was equal to that of the normal airplane with slats operating.

High-Speed Longitudinal Stability

At Mach numbers between 0.75 and 0.94, the maneuverability of the normal airplane is limited by a longitudinal instability which makes it difficult to obtain high normal accelerations without inadvertently "overshooting" or pitching up to a stall. Reference 3 has shown that this longitudinal instability is due primarily to an unstable break in the wing-fuselage pitching-moment coefficient caused by a loss of lift over the outer portions of the wing from shock-induced separation. In addition, there are trim changes with Mach number in which the airplane tends to pitch up at 0.95 Mach number when slowing down from a high Mach number dive.

Since these high-speed longitudinal-stability problems are associated with shock-induced separation, it was not expected that the modified leading edge, which was designed to improve the low-speed maximum lift coefficient, would reduce the high-speed longitudinal instability. It was hoped, however, that the amount of camber used would be insufficient to adversely affect it, as was shown to be the case in reference 4 for tests at 2,000,000 Reynolds number of a wing with a similar leading-edge modification.

Flight tests confirmed that the high-speed longitudinal instability was little affected by use of the modified leading edge. In figure 8, the static longitudinal stability as shown by the variation of δ_e with A_Z for the modified leading edge compared closely with that for the normal leading edge. It would follow that the stick-force variation with A_Z would be similar to that for the normal wing.

In regard to trim changes with Mach number, figure 9 shows the elevator angle required for level flight in the Mach number range from 0.60 to 0.91 for both configurations. Here, too, the addition of the modified leading edge caused no significant change.

Above 0.91 Mach number, data are not available on trim changes with change in speed, but on the basis of four dives through this speed range the pilot reported that no significant change resulted from the installation of the modified leading edge. The pitch-up was still apparent when slowing down through 0.95 Mach number and was of approximately the same intensity.

Although no specific tests were made for buffeting, the pilot felt that the modified leading edge did not alter the buffeting characteristics from those shown in reference 5 for the normal airplane.

Figure 10 presents the variation of the flight measured wing-fuselage pitching-moment coefficient with normal-force coefficient at various constant values of Mach number for both the modified and normal leading edge. These data agree quite well with the results obtained in reference 4 at a Reynolds number of 2,000,000. Below 0.8 Mach number there was no significant change resulting from the use of the modified leading edge, only a negative shift in pitching-moment coefficient at zero lift of -0.01, the variation with C_N being almost identical for the two configurations. The increased elevator angle required with the cambered leading edge as shown in figures 8 and 9 is about a half a degree smaller than would be expected from the C_{m_0} shift shown in figure 10. Above 0.8 Mach number there is, in addition, a small increase in stability below the C_N for pitch-up with the break coming at slightly lower C_N.

Drag

The drag of the airplane (shown in fig. 11) was measured using the technique described in reference 5. The drag was measured at normal-force coefficients between 0.12 and 0.18 and was corrected to $C_N = 0.15$ using a value of 0.6 for the airplane efficiency factor at all Mach numbers.

The drag of the modified airplane appeared to be slightly higher than that of the normal airplane up to about 0.86 Mach number. The data of reference 4 tend to substantiate this result showing a higher minimum drag for the wing with forward camber. It should be noted that in the referenced results this drag penalty disappeared at a lift coefficient of 0.3 and, at higher C_L values, the drag of the basic wing was higher.

For Mach numbers between 0.86 and 0.94 the drag of the two airplane configurations appeared to be equal. Above 0.94, the drag of the modified wing began to increase significantly above that for the basic wing.

CONCLUSIONS

Flight tests of a 35° swept-back-wing airplane, with the wing modified to incorporate a moderate amount of camber over the forward portion of the chord and an increase in the leading-edge radius, have shown the following:

1. The modified leading edge provided an increase in maximum lift coefficient, flaps down, of 0.31 over the basic wing and of 0.22 over the wing with slats operating. The stalling characteristics in steady straight flight were considered unacceptable by the pilot because of an abrupt roll-off. In addition, there was no stall warning. The installation of a fence resulted in a satisfactory stall with a penalty in maximum lift of about 11 percent. Thus the maximum lift characteristics compared closely with those for the normal airplane with slats operating. Longitudinal instability beyond the stall noted in wind-tunnel tests was not apparent to the pilot.

2. The addition of the cambered leading edge caused no significant changes in the longitudinal-stability characteristics of the airplane in tests which extended to a Mach number of 0.91. Wind-tunnel tests of a similar wing at 2,000,000 Reynolds number indicated similar results.

3. The drag of the airplane with the modified leading edge was slightly higher than that of the normal airplane below 0.86 Mach number and above 0.94 Mach number at 0.15 normal-force coefficient.

National Advisory Committee for Aeronautics
Ames Aeronautical Laboratory
Moffett Field, California

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3. Anderson, Seth B., and Bray, Richard S.: A Flight Evaluation of the Longitudinal Stability Characteristics Associated With the Pitch-Up of a Swept-Wing Airplane in Maneuvering Flight at Transonic Speeds. NACA RM A51I12, 1951.

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TABLE I.- DESCRIPTION OF TEST AIRPLANE

Wing	
Total wing area (including flaps, slats, and 49.92 sq ft covered by fuselage)	287.90 sq ft
Span	37.12 ft
Aspect ratio	4.79
Taper ratio	0.51
Mean aerodynamic chord (wing station 98.7 in.)	8.08 ft
Dihedral angle	3.0°
Sweepback of 0.25-chord line	35°14'
Sweepback of leading edge	37°44'
Aerodynamic and geometric twist	2.0°
Root airfoil section (normal to 0.25-chord line)	NACA 0012-64 (modified)
Tip airfoil section (normal to 0.25-chord line)	NACA 0011-64 (modified)
Ailerons	
Total area	37.20 sq ft
Span	9.18 ft
Chord (average)	2.03 ft
Horizontal tail	
Total area (including 1.20 sq ft covered by vertical tail)	34.99 sq ft
Span	12.75 ft
Aspect ratio	4.65
Taper ratio	0.45
Dihedral angle	10.0°
Root chord (horizontal-tail station 0)	3.79 ft
Tip chord, equivalent (horizontal-tail station 76.68 in.)	1.74 ft
Mean aerodynamic chord (horizontal-tail station 33.54 in.)	2.89 ft
Sweepback of 0.25-chord line	34°35'
Airfoil section (parallel to center line)	NACA 0010-64
Maximum stabilizer deflection	+1° up, -10° down
Elevator	
Area (including tabs and excluding balance area forward of hinge line)	10.13 sq ft
Span, each	5.77 ft
Chord, inboard (equivalent horizontal-tail station 6.92 in.)	1.19 ft
Chord, outboard (theoretical, horizontal-tail station 76.18 in.)	0.57 ft
Maximum elevator deflection	35° up, 17.5° down
Boost	Hydraulic

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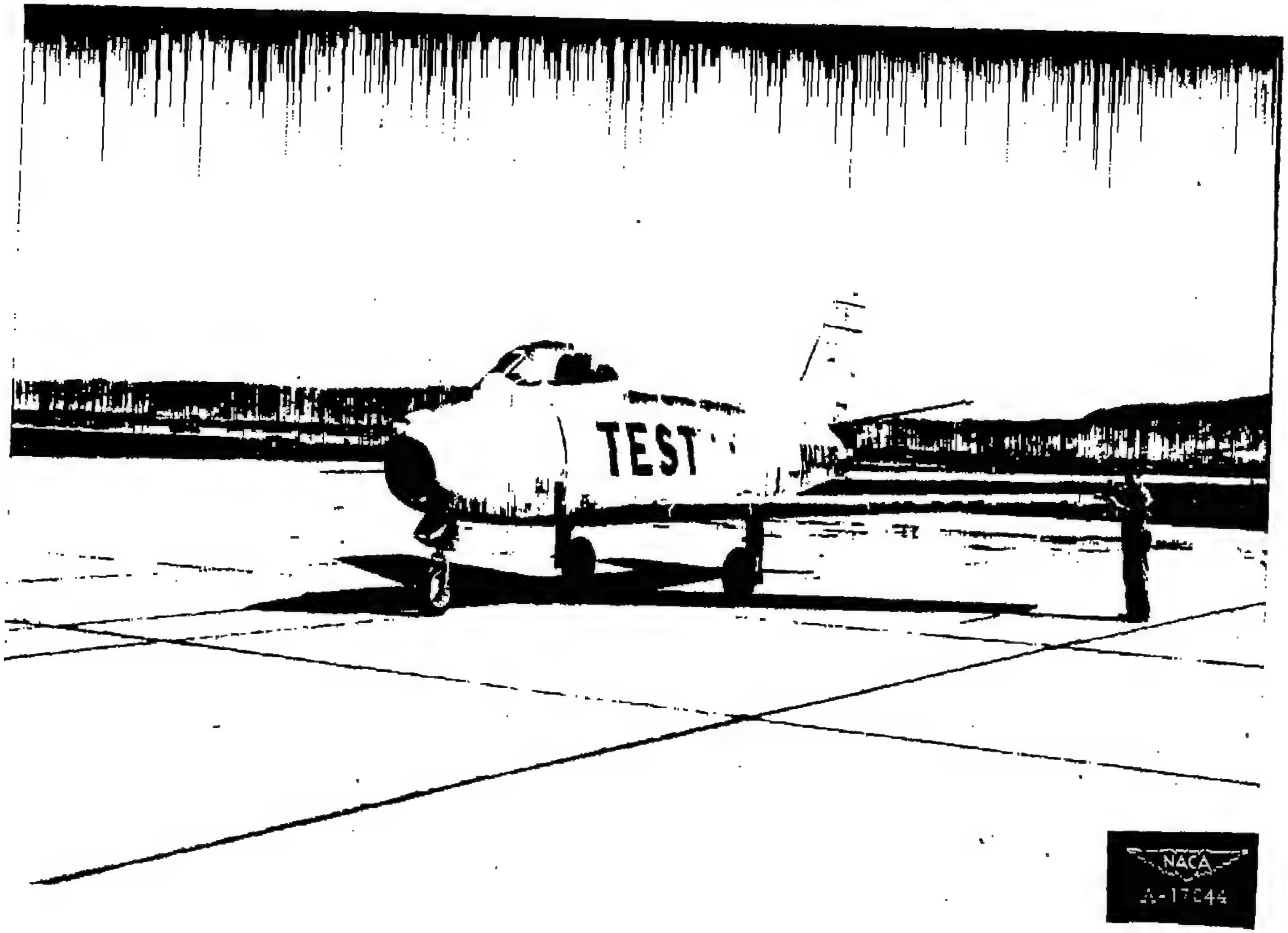


Figure 1.– Three-quarter front view of test airplane.

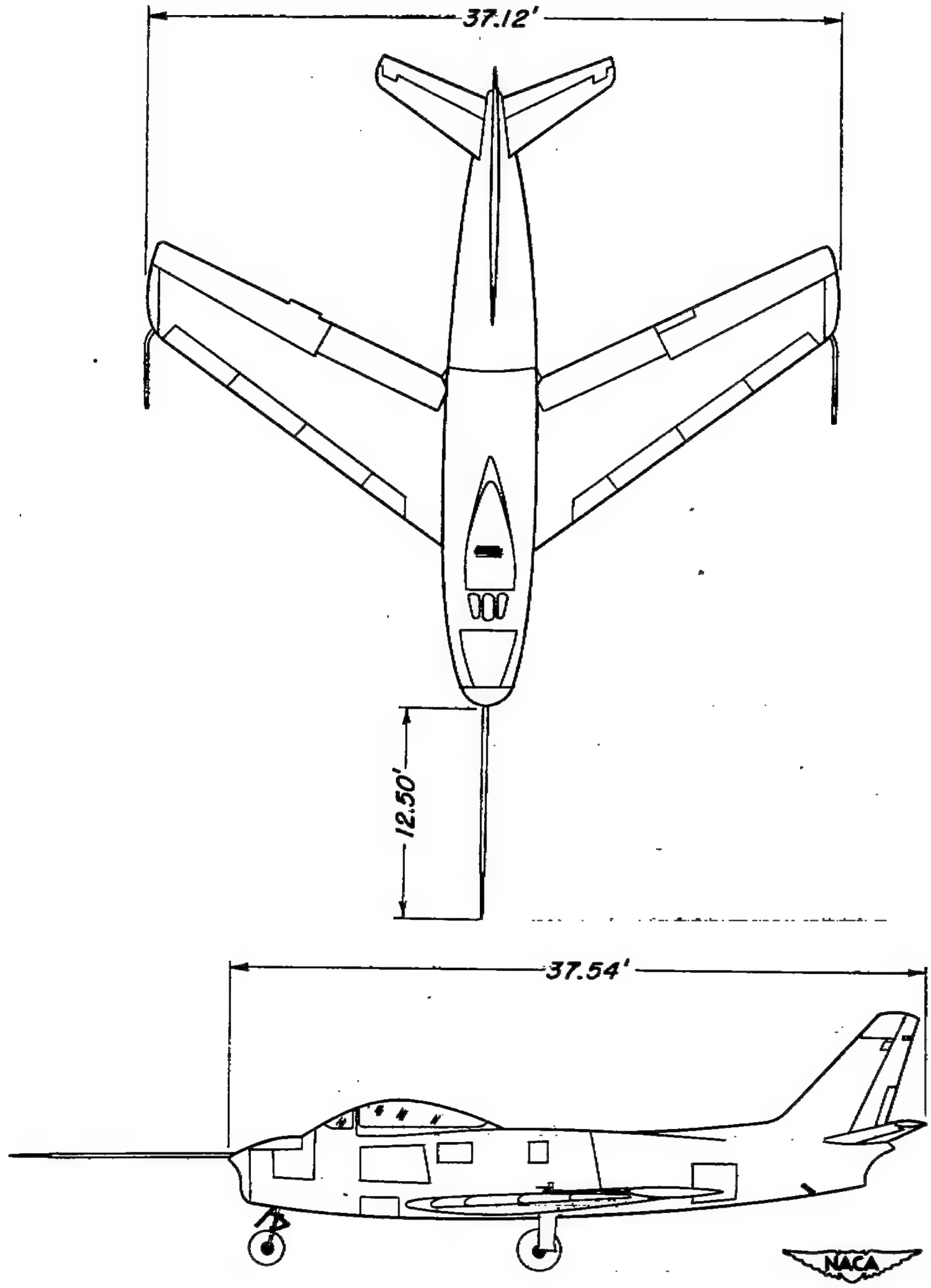
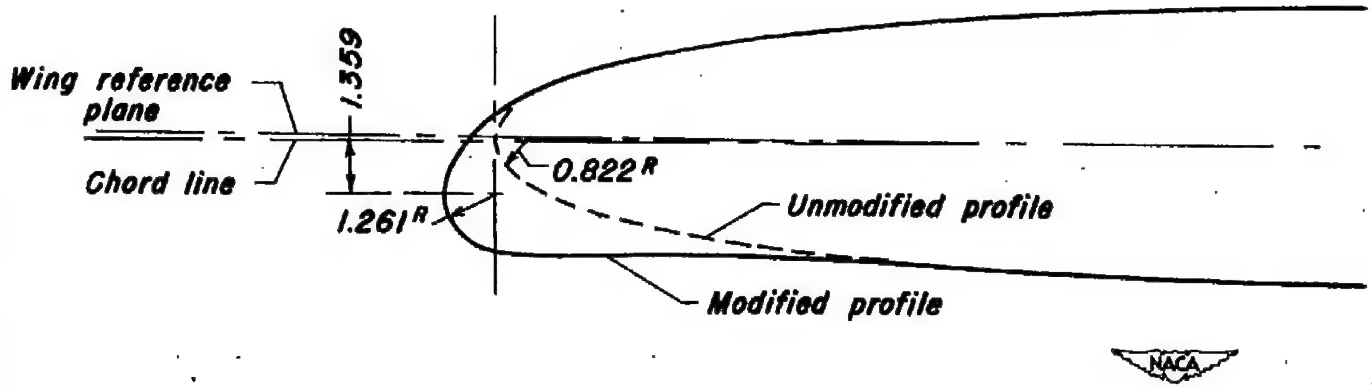


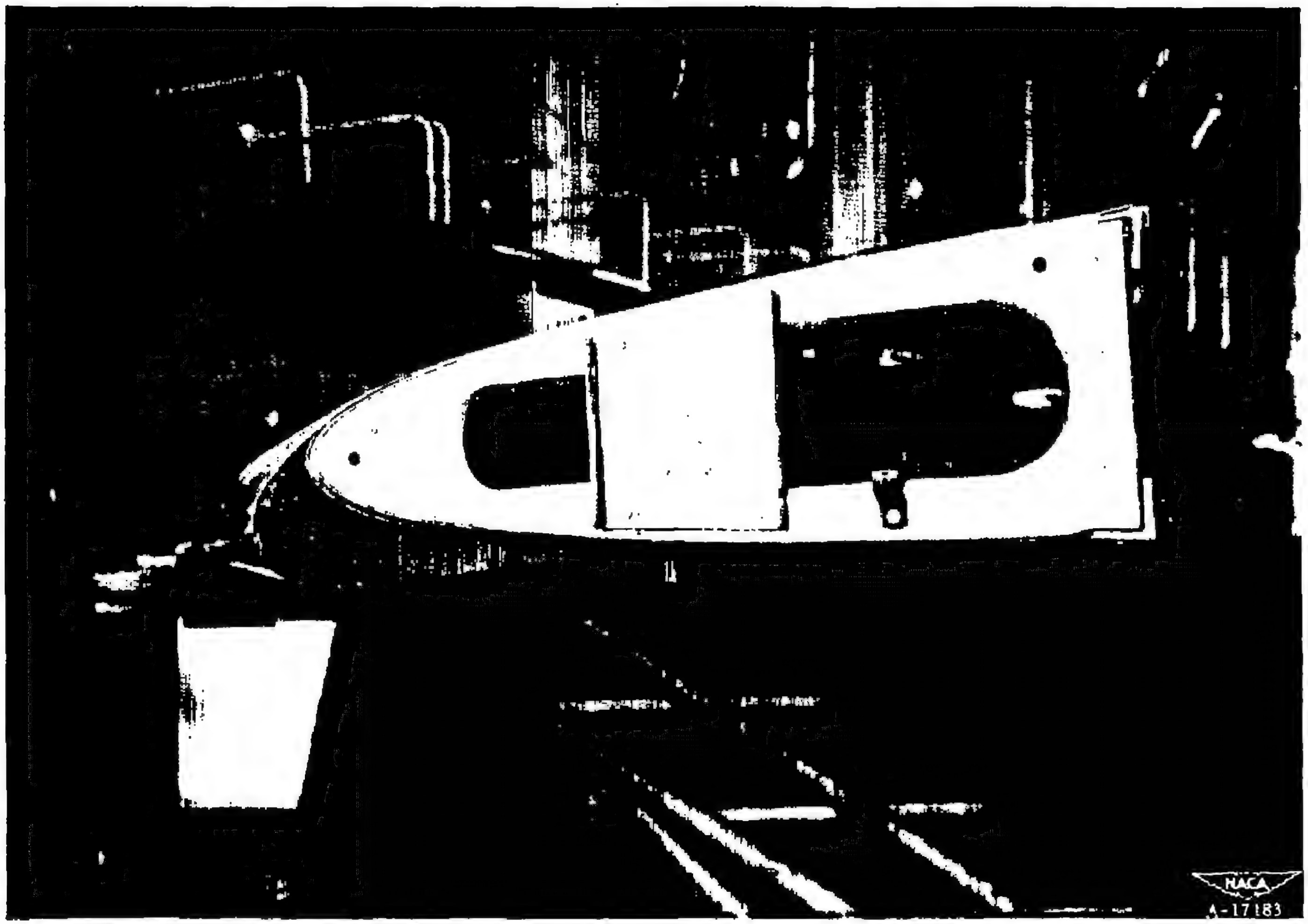
Figure 2.- Two-view drawing of the test airplane.

Note: All dimensions are in inches.



(a) Details of airfoil sections at 0.857 semispan, taken normal to the wing quarter-chord line.

Figure 3.- Unmodified and modified wing airfoil sections.



(b) Photograph of wing leading-edge modification.

Figure 3.– Concluded.

Figure 4.– View of 0.25c fence at 0.63 semispan.

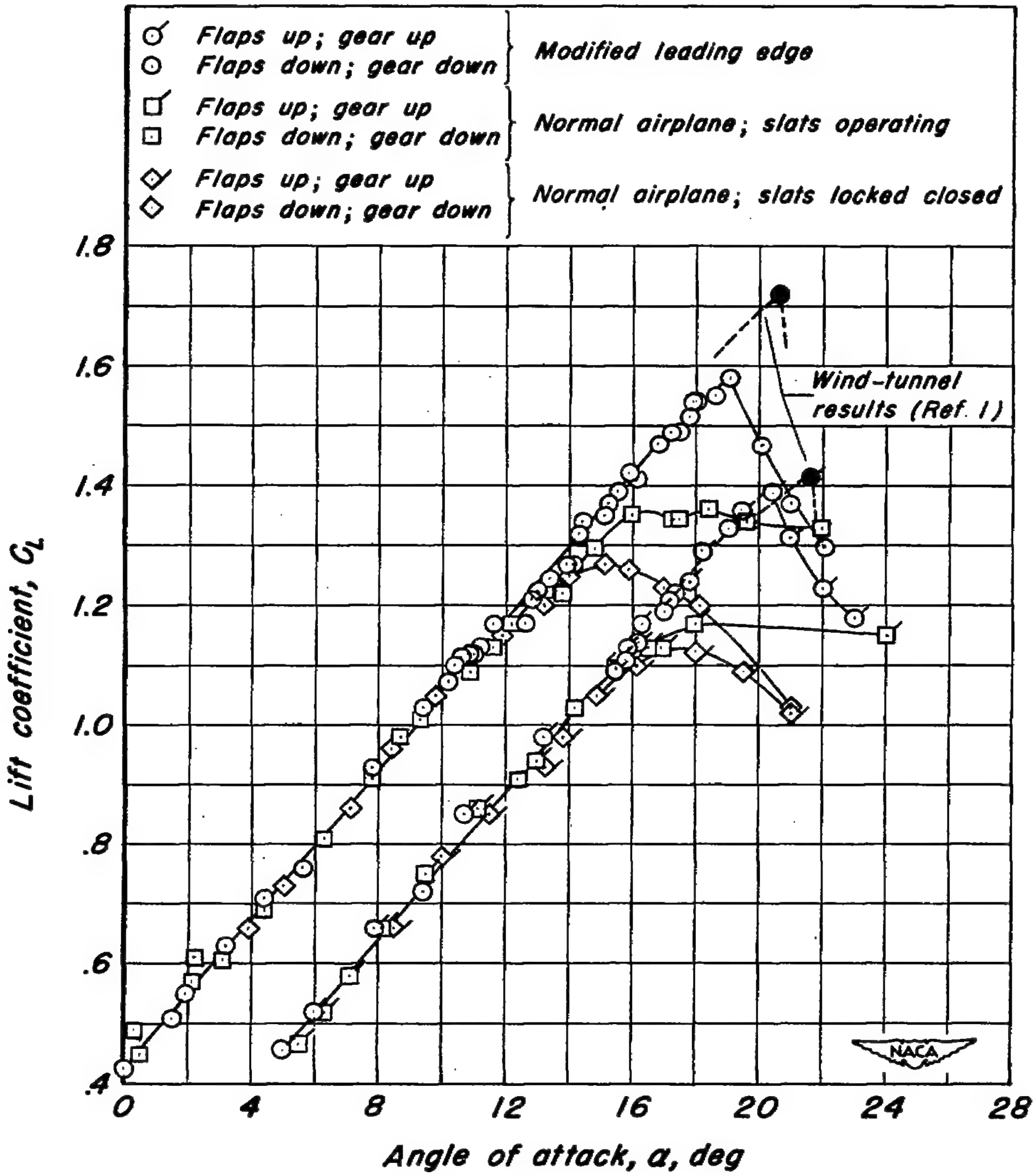


Figure 5.—Variation of lift coefficient with angle of attack for various configurations.

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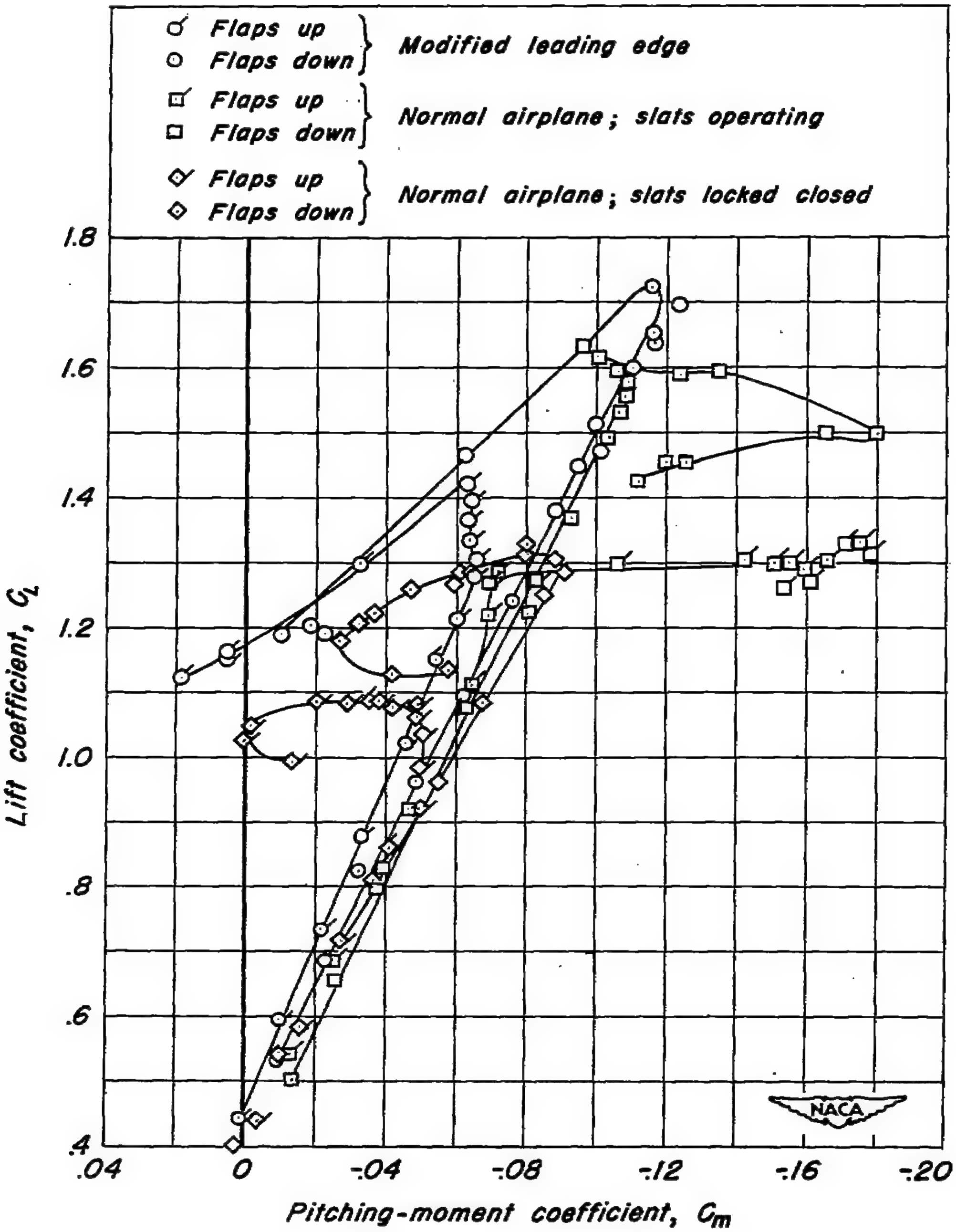

Figure 6.– Variation of pitching moment with lift coefficient as obtained from wind-tunnel tests (reference 1).

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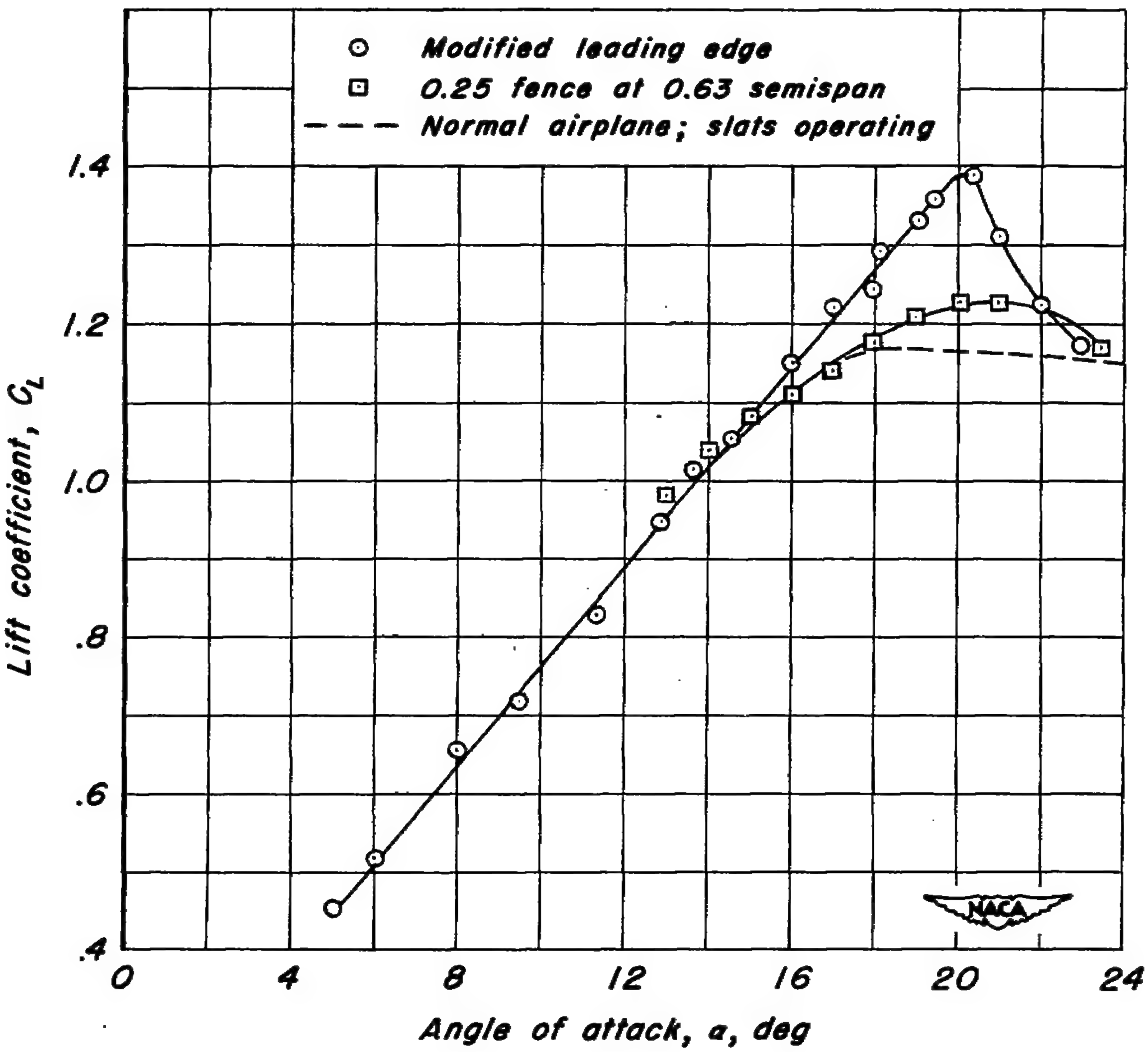

(a) Flaps up, gear up.

Figure 7.— Comparison of various wing modifications on the lift curves.

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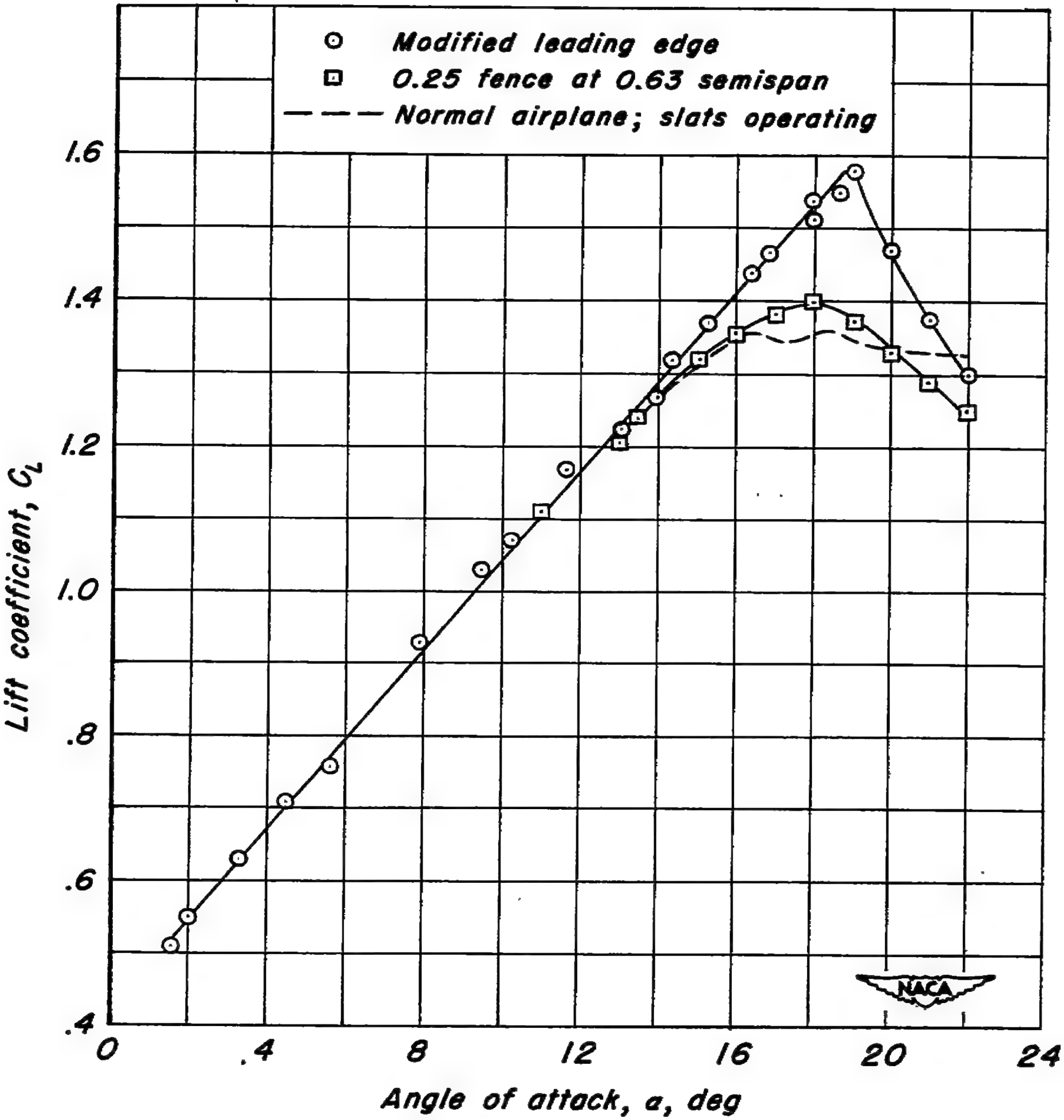

(b) Flaps down, gear down.

Figure 7.— Concluded.

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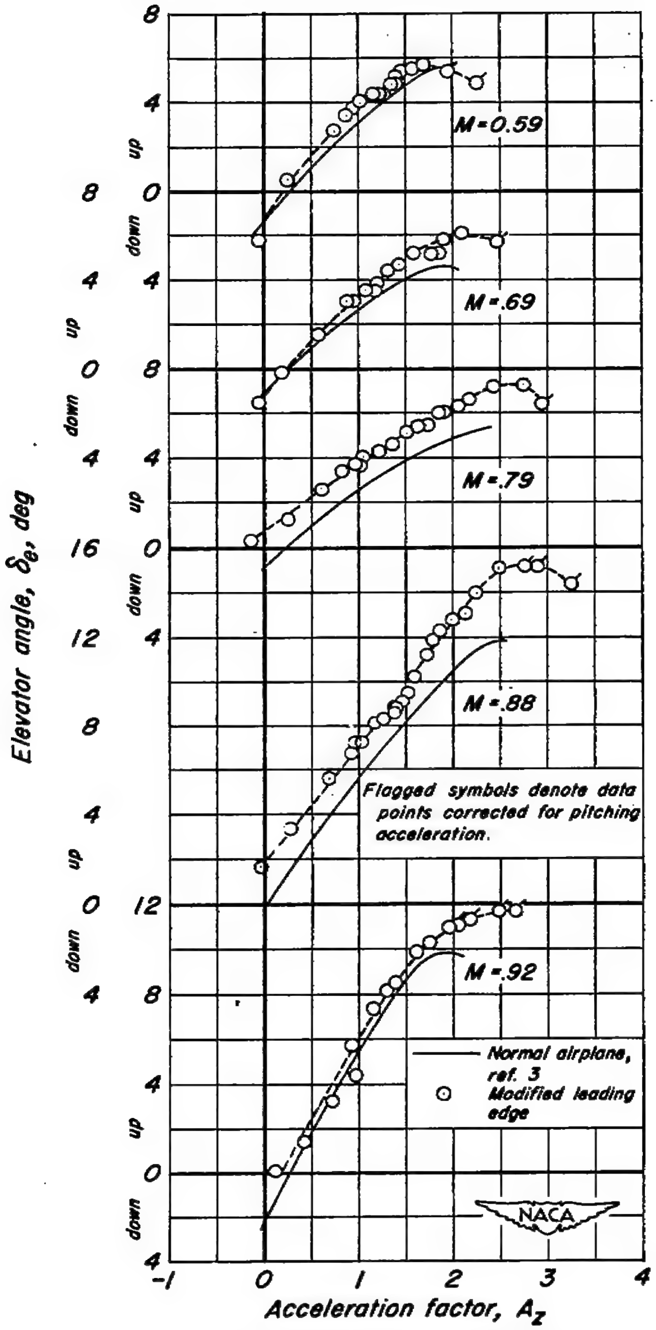

Figure 8.— Variation of elevator angle with normal acceleration factor, A_z, for various Mach numbers.

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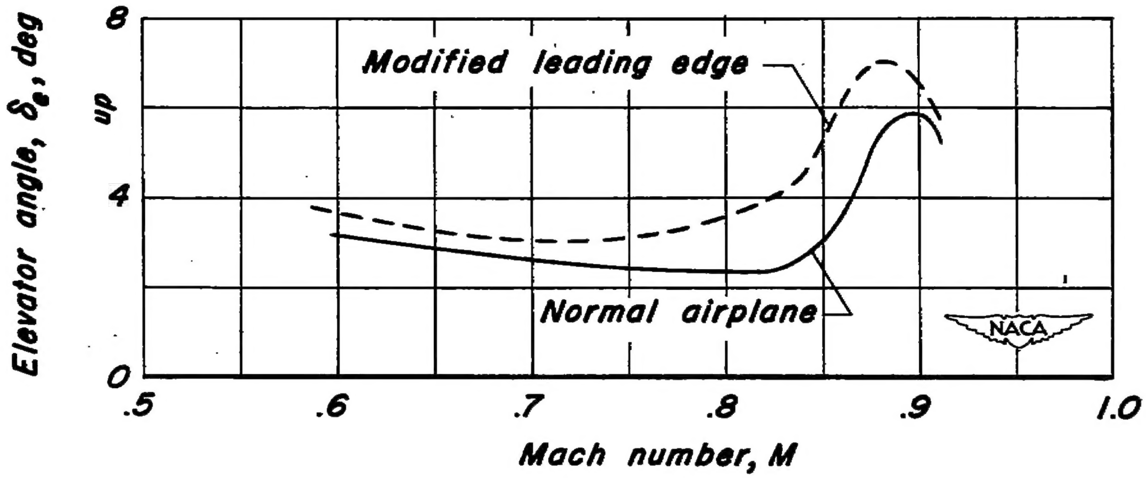

Figure 9.– Variation of elevator angle for steady straight flight. Stabilizer incidence, 0.6°.

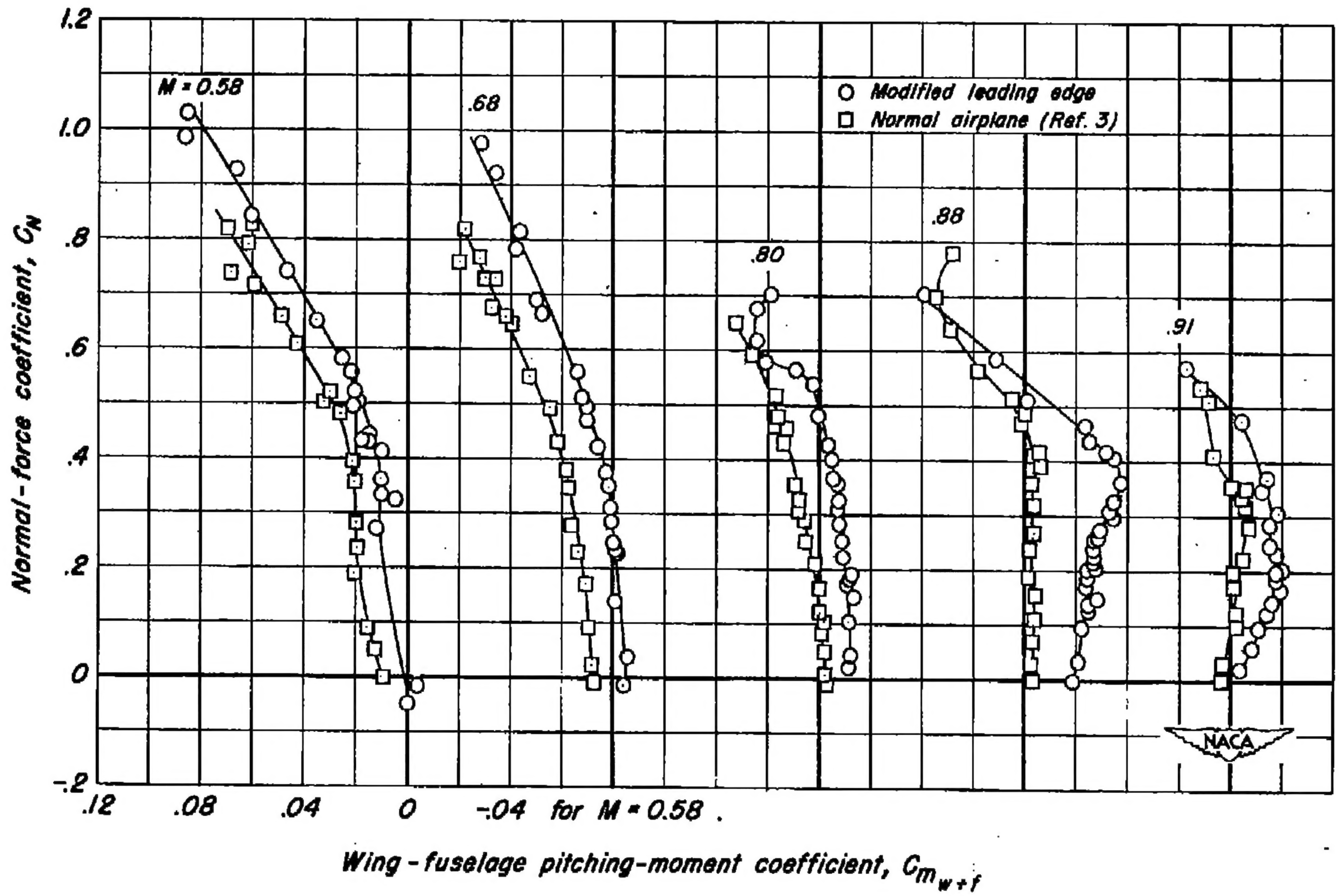

Figure 10.– Wing-fuselage pitching-moment coefficient of airplane with normal wing and with modified leading edge.

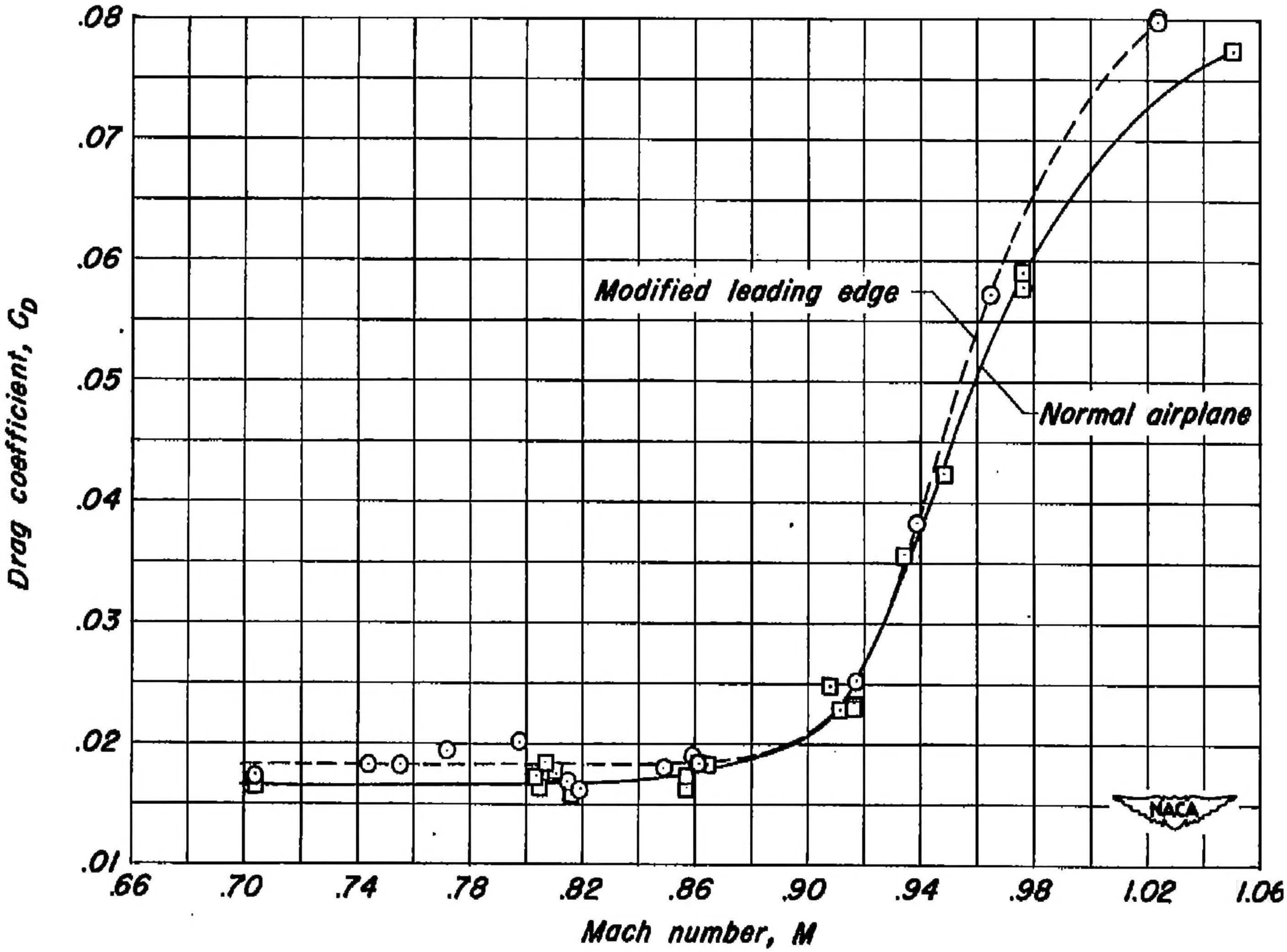

Figure 11.– Variation of drag coefficient with Mach number for measured points between C_N values of 0.12 and 0.18 corrected to a C_N of 0.15.

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